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EXPERIMENTAL INVESTIGATION OF ROCKET PLUME RADIATION AT LOW ALTITUDES

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Calspan Corporation

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This report is a midterm technical report on a research program currently underway at Calspan on the experimental investigation of rocket plume radiation at low altitudes. The objectives of the program include the study of plume structure, gasdynamics and chemistry associated with the interaction of the plume with the ambient atmosphere. The experimental facilities, techniques and diagnostics are described in detail. The overall system as applied to this program is presently operational and undergoing initial checkout tests. The overall test parameter

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SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered) matrix in terms of rocket motor configuration, propellant mixtures, altitudes and ambient gas conditions is presented; the tests will continue through July, and a final report on this phase of the work is scheduled for 15 August 1974.

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Calspan

MIDTERM TECHNICAL REPORT

EXPERIMENTAL INVESTIGATION OF ROCKET PLUME RADIATION AT LOW ALTITUDES

W.H. Wurster and D.W. Boyer

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SECTION 1

INTRODUCTION

The objective of this research program is to obtain laboratory data pertinent to low-altitude rocket plumes. At its inception, diagnostic measurements were to be made of rocket test firings conducted in the Calspan Ludwieg tube (L-T) facility, which permits the testing of rockets in supersonic external flows. The primary thrust of these measurements is directed toward the determination of the gasdynamics and structure of amine-type rocket plumes for a broad range of test parameters. These parameters include altitude, thrust, rocket nozzle area ratio, nozzle clustering, and propellant O/F ratio. Baseline external flows of nitrogen will be used; selected runs with air will assess the influence of afterburning. The instrumentation includes a short wave infrared (SWIR) scanning spectrometer for species identification and relative concentration measurements, and a multichannel spatial-scanning radiometer for direct measurements of axial and radial radiance profiles of the plume at selected wavelengths. Total temperature probes and camera coverage of the plumes will also be implemented.

A two-month testing period was planned in the L-T facility, which is NASA-owned, and used with permission on a non-interference basis.

Recently, however, it has become doubtful whether the two-month test interval would be available during the contract performance period, because of upgraded commitments to tests conducted on the NASA Space Shuttle Program.

Amendments to the Calspan test program were thus negotiated which permitted the testing schedule to continue unbroken, while preserving the original objectives of the overall ARPA program on low-altitude plume studies. Thus, tests will be started in the Calspan altitude test chamber.

The thrust of these tests is basically plume-chemistry oriented, with measurements to be made of the chemical aspects of ignition and quenching in the afterburning region of the plume as a function of altitude, but without external flow. The instrumentation is the same as that cited above, and many of the test parameters such as the altitude and rocket motor characteristics are basically the same.

It has been proposed to complete the L-T tests during FY 75 in a series of tests that can take full advantage of the findings of the present program. This proposal has been technically approved by ARPA, and will permit an integrated, cost-effective approach to both the chemical (altitude chamber) and plume structure (Ludwieg tube) aspects of the program to be made.

The overall program is structured to be flexible and responsive to inputs derived from plume analysts and modellers and from field data. As such, technical liaison is being maintained with Dr. H. Wolfhard of IDA, Drs. F. Simmons and J. D. Stewart of Aerospace, Dr. R. Vaglio-Laurin of Advanced Technology Labs., Dr. J. Draper of Aerodyne Research, and Dr. F. Boynton of Physical Dynamics.

At the present time, rocket motor check-out firings are being conducted. For these tests an existing motor was modified for operation at higher combustion chamber pressures. A complete description of the motor configuration, its operation and the use of the gaseous equivalent propellant technique is presented in Section 4 of this report. It is pointed out that motor operation can be initiated by spark-plug ignition, which permits repeated testing without facility shut-down for diaphragm replacement, as is usually the case with short-duration test techniques. To take advantage of this feature, a new motor support system was designed and implemented, which permits large motor excursions (relative to the fixed optics) to be made remotely. In addition, the optical diagnostics locations were changed to permit plume measurements to be made well downstream of the rocket-motor exit plane.

Section 2 of this report describes the altitude chamber and the impulsive rocket testing techniques employed at Calspan. The diagnostic instrumentation used in these experiments is discussed in Section 3. The design and operating criteria for the rocket motor is discussed in Section 4, and the test parameter matrix and current program status are summarized in Section 5.

SECTION 2

CALSPAN ALTITUDE CHAMBER AND TEST TECHNIQUES

The altitude test facility consists of a 10-ft. diameter by 28-ft. long chamber which is capable of being evacuated to pressures of the order of 5×10^{-5} torr by means of a large, refrigerated chevron-baffled oil diffusion pump. The chamber has been extensively employed in previous studies of exhaust-plume radiance performed under ambient altitude conditions ranging from 40,000 to 350,000 feet. The chamber is equipped with a number of ports to which a variety of diagnostic instruments can be attached. Figure 1 depicts a schematic diagram of the configuration relevant to this test program. As shown, the rocket motor is positioned (relative to the fixed fields-of-view of the instruments) near one end of the tank.

As will be discussed in Section 4, the rocket motors are designed to be used in an impulsive, or short-duration run mode. In this manner, high thrust configurations can be readily accommodated. The basis of the impulsive test technique lies in the fact that ambient and rocket flow conditions need be maintained only long enough to establish the plume flow field and obtain the desired measurements, see for example, Refs. 1-3. The available test time in the altitude chamber is determined by the time required for the plume gas to reach the end of the chamber and propagate a disturbance back to the motor station (typically 20-40 ms). That is, until the fact that the test chamber is of finite size is communicated back to the test region (by gasdynamic wave processes), the exhaust plume flow field behaves as if it were operating in an infinitely large test volume. Since the actual test altitude is that corresponding to the ambient pressure in the vacuum chamber immediately preceding rocket firing, continuous pumping of the rocket exhaust products is not necessary to maintain test altitude during the short test time; thus, a vacuum pumping capability adequate to bring the tank to the desired test altitude in a reasonable time is all that is required. It is noted that a modification of this procedure will be employed in this test program in order to satisfy ambient gas impurity-level criteria. The tank will be pumped out after each run and filled with a pure O_2/N_2 (simulated-air) mixture. This point is discussed further in Section 4.

SECTION 3

DIAGNOSTIC INSTRUMENTATION

The instrumentation to be utilized during the proposed program have been developed and tested at Calspan during previous plume investigations.[4] These include a rapid-scanning SWIR spectrometer which permits absolute spectral radiances over the 2-8 μm wavelength range to be measured during the motor burn. A second instrument that has recently been implemented is a spatially scanning, multi-wavelength radiometer, capable of producing axial and radial radiance profiles of the plumes. In addition, for these studies, high quality photographic data will be obtained, both colored and with selected spectral definition. Finally, a set of probes will be used to obtain total temperatures throughout the plume to aid in temperature definition. Each of these diagnostics will be briefly described below.

A diagram of the proposed overall experimental configuration is presented in Figure 1. The fields of view of the instruments are indicated, together with the range of motion depicted to illustrate the total geometric range to be covered. Data can be taken to at least six feet downstream of the exit plane. Also shown is the location of a radiation source on the floor of the chamber. Its purpose is two-fold: it supplies a fiducial mark in the spatial scanner records. Also, by means of an externally-controlled movable mirror, it serves as a source in the SWIR spectrometer field of view. In this way, the level of residual H_2O or CO_2 in the ambient test gas may be monitored prior to each run.

An illustration of the typical plume structure and definition obtained in the Calspan facility is shown in Figure 2, taken from Reference 4. As can be seen, typical plume features are well defined and amenable to quantitative measurements.

3.1 SWIR Rapid-Scan Spectrometer

The mode of operation of this instrument is shown in Figure 3. It will be used to obtain absolute spectral radiance at selected stations in the plumes. The nature of the spectra are depicted in the figure. The instrument employs a circular variable filter (CVF) wheel rotating in front of two detectors (See Figure 4). The CVF is spun at 8000 rpm by a 400 Hz synchronous motor, and is made in two semicircular segments spanning the wavelength ranges 2-4 and 4-8 μm. One detector is InSb, covering the range between 2 and 5.5 μm. The balance of the spectrum is recorded by the second detector which is Ge:Au. This arrangement thus permits the entire spectrum to be scanned in 7.5 msec. Since a nominal rocket test time is on the order of 30 msec, this readily permits several spectra to be recorded during this interval.

The detector is imaged onto the rocket plume axis by means of a silicon lens. The detector area is 1/2 x 2 mm, and with a magnification factor of approximately 3, the plume area subtended by the detector is about 1.5 x 6 mm. By means of a rotary-solenoid actuated mirror, the SWIR system was calibrated by ready reference to a standard black body source whose response at the detector, in terms of a similar radiation source located on the plume centerline, was known. The procedures for irradiance and wavelength calibration as well as the data reduction techniques are detailed in Reference 4.

In operation, the rotating CVF inherently produces a modulated background signal, upon which the plume radiance signals are superimposed. In order to optimize the dynamic range of the SWIR scanner, this modulated background signal from the CVF is electronically compensated. This is achieved by means of a 10-channel ramp-generating network in which the background signal waveform is closely matched by straight-line segments. Adjustments for small variations in slope and duration are provided for each line segment. The compensated, or difference signal then represents the reference SWIR background, or baseline signal with respect to which the plume radiance signal is measured during a run. The method has proved to be consistent and reliable since only minor amplitude adjustments are required from run to run to maintain the desired background-signal compensation.

A photodiode pickup system is coupled to the rotating CVF by means of six peripheral tabs on the wheel. These tabs provided fiducial wavelength signals, as well as synchronization capability, such that injection of the compensating signal occurs consistently at the 2 μm position in the scan.

Figure 5 shows a typical spectrum obtained with this instrument of the plume radiance from a rocket motor using a conical nozzle and N_2O_4/MMH propellant. As expected with this propellant, H_2O at 2.7 μm and CO_2 at 4.3 μm are the dominant radiators. The two spectra shown are from duplicate tests. They are indicative, therefore, of the overall reproducibility including rocket motor performance, data acquisition, reduction, and calibration.

Relocation of this instrument for the current tests has been completed and a new CVF drive motor has been installed (necessitated by a bearing failure). The complete system is undergoing final calibration at the present time.

3.2 Multi-channel, Spatial-Scanning Radiometer

This instrument was recently developed under ARPA sponsorship[4] and has also proven to provide significant diagnostic data on rocket plumes. It is particularly useful in determining both axial and radial radiance profiles, and will be used in the present program to provide radiance maps of the plume flowfield at selected wavelengths.

Heretofore, diagnostic optical instruments have either been wavelength-bandpass-filtered radiometers or spectrally-scanned instruments, l .ited to obtaining data along fixed lines of sight through the test plumes. The dependence of the radiance as a function of position throughout the plume was obtained by changing the position of the rocket nozzle relative to the FOV for a restricted number of tests. This instrument was designed to overcome this restriction by performing a spatial scan through the plume during the test-time interval of the rocket firing. In this way it supplements the fixed FOV instruments, and at the same time provides direct and continuous data on radiation gradients in the plume. It uses an InSb detector array, which permits separate wavelength intervals between 1-5 μm to be selected for measurement. Thus, for example, CO_2 at 4.3 μm, H_2O at 2.7 μm and continuum radiance at 3.8 μm can be simultaneously measured. The instrument is shown schematically in Figure 6, and essentially consists of a rotating plane mirror which is oriented to scan along a particular direction in the plume during a portion of its 360° rotation.

As depicted in Figure 6 the radiometer is in the axial-scan mode. However, the instrument may readily be rotated through 90° to provide for radial scans across the plume. Furthermore, the mirror and drive motor assembly are mounted on a platform which is provided with a tilt adjustment. Off-centerline axial scans may therefore be performed as well as transverse plume scans at selected axial locations along the plume. The mirror is rotated at 225 RPM by a 60 Hz hysteresis synchronous planetary gearmotor employing a 16:1 speed reduction. A photodiode pickup attached to the mirror shaft generates a pulse once in every 270 msec revolution period of the spatial-scanner mirror. This pulse then represents the t = 0 trigger pulse which initiates the motor operation sequence and the other diagnostic instrumentation. The angular displacement between the photodiode pickup and the plane mirror is usually set so that the axial scan sweeps by the nozzle exit plane location immediately following the establishment of steady rocket-motor operation.

This instrument has, to date, only been used in the axial scan mode. The scan length along the nozzle centerline was that afforded by the field of view through a 2 x 7-inch Irtran window in the base of the instrument. The axial scan was imaged onto the array of InSb detectors by means of an 8-inch focal length (at 4 μm) Irtran 2 lens. Narrow-band filters were installed, as required, in front of the nine-detector array for radiation-gradient measurements at 2.7 μm and 4.3 μm. Only two of the nine InSb detectors were employed for plume-radiance monitoring to date.

To be compatible with the short-pulse rocket motor test technique, the optical geometry of the system was designed such that a 48-inch axial centerline scan was performed in 16 msecs; with a magnification factor of about 8, the 0.5 x 1 mm detectors subtended an area at the plume centerline of about 4 x 8 mm.

The lower record in Figure 2 shows a typical scan obtained with this instrument, for the conditions listed in the figure. The 4.3 μm infrared centerline plume structure is seen to be clearly resolved, and to correspond with the internal shock structure of the plume shown in the simultaneously recorded photograph.

For the present tests, the instrument has been modified in several ways. First, provisions have been made to rotate the mirror remotely, to any fixed angular displacement. For operation in a stationary mode, this feature permits recording the plume radiance as a function of time at any point of a very broad extent of plume volume. Secondly, an Irtran window was installed in the cover of the instrument and provision made for mounting a shuttered blackbody standard radiation source directly over the window. This arrangement will permit direct intensity calibrations to be made, and will be especially useful for obtaining the relative response functions for each of the detectors in the array used for simultaneous plume radiance mapping. Each of these modifications have been completed, and the instrument is ready for use as described.

Finally, a modification to the optical train has been designed and is presently being implemented. When completed, it will be incorporated into the instrument. This modification effectively permits detector image rotation through 90° and permits the plume to be radially traversed at several axial positions simultaneously. This feature will greatly facilitate plume radiance mapping.

3.3 Probe Measurements

A probe has been constructed for the measurement of total temperatures in the plume. The probe is essentially a shielded thermocouple which senses the temperature of the gas in a vented cavity. (See Figure 7). The probe is based on a design described by Winkler[5] for temperature profile measurements in supersonic boundary layers.

A similar probe design has also been employed by Holden,[6] in a shock tunnel at Calspan, for turbulent boundary layer and wake-flow total-temperature measurements. These latter probes, which were operated as resistance - thermometers, were of small scale (.150 x .050 in. inlet) patterned after those described by Weinstein.[7] The requirement on probe response time in the shock-tunnel flows, which necessitated the very fine-wire, resistance-thermometer approach by Holden, is relaxed in the case of the present plume experiments. The plume flow durations will be of order 30 msecs, or almost an order of magnitude longer than the shock-tunnel test time. As a result, the larger thermocouple type of sensor can be used, with adequate margin, in spite of its longer response time. The advantage is then one of simplicity of temperature interpretation, improved ruggedness, and less sensitivity to conduction losses (or length/diameter effects) since the sensing element is the junction at the wire center.

Based upon the probe-response studies of Winkler,[5] the vent area to inlet area for the probe was selected as 0.2. The probe inlet size is .125 x .25 in. The thermocouple is iridium/iridum 40% rhodium of wire thickness 0.0006 in. Iridium was employed because of its improved resistance (over tungsten) to oxidation effects in hot plume flows containing CO and H_2O. The thermocouple EMF - temperature data were obtained from the NBS data by Olsen.[8]

A heat-transfer gauge has also been installed in a 1/2-inch diameter rod which will be positioned in the plume for the measurement of local stagnation-region heat-transfer rates.

3.4 Ancillary Diagnostic Measurements

3.4.1 Photographic Coverage

Pictures such as that shown in Figure 2 have been found to be especially useful in the visualization, definition and location of plume-structure features. Such pictures, taken with ASA 3000 speed Polaroid film are readily and routinely obtained, requiring only a synchronized shutter to permit exposure only during the steady-state portion of the motor burn.

Provisions have been made to attempt quantitative color photographs with an f/4 wide angle Hasselblad camera. If suitable exposures are obtainable, it is also planned to use filtered photographs in order to obtain quantitative spatial data to aid in elucidating the source of the visible plume radiance under the range of program test parameters.

3.4.2 Fixed-FOV Radiometer Measurements

As a low-priority additional diagnostic, a simple radiometer (multiplier phototube, filters, apertures) has been installed to record any plume radiance at wavelengths in the range between 2500-4000 Å. In particular, narrow-band filters, both in and just outside the 3064 Å band-head of OH will be employed. Although not expected from equilibrium considerations at the temperatures involved (all less than $\sim$ 3000°K), any appreciable chemiluminescence will be recorded.

3.4.3 Housekeeping Functions

During each test, records are routinely made of the motor fuel and oxidizer pressures, and the combustion chamber pressure as a function of time throughout the motor burn. In addition, the time of the ignition pulse and all subsequent event synchronizations are recorded, to ensure that all data were obtained during steady-state rocket motor operation.

SECTION 4

ROCKET MOTOR: DESIGN CRITERIA AND OPERATIONAL TECHNIQUES

The initial plume - chemistry experiments will be performed in the large altitude chamber under static ambient conditions (no external flow), corresponding to equivalent density altitudes from 5 to 30 Km. The rocket motor will be the R-4D, which was employed in the preceding ARPA high-altitude plume program.[4] The nozzle is conical and the first experiments will be performed with a nozzle exit-plane area ratio $\epsilon = 6.8$, using the amine propellant combination N_2O_4/MMH. Rocket motor operation will employ the gaseous-simulation technique which has been used in previous plume-radiance investigations (e.g. Ref. 4), and which is discussed below.

An overall schematic of the fuel/oxidizer system is shown in Fig. 8. A photograph of this motor with two nozzle configurations is shown in Figure 9. Preparatory to the experiments, program investigations thus far have comprised the specification of the requirements for motor thrust level, fuel and oxidizer mass flow rates, injector operation and propellant venturi sizing. In addition, the fuel and oxidizer requirements for motor operation at arbitrary equivalent O/F_{Liquid} were established. These investigations are briefly documented herein.

4.1 Thrust Level

The initial experiments will be performed at a nominal motor thrust level of 500 lbs. The motor thrust is given by

$$F = C_F A_t p_c$$

where p_c is the combustion chamber pressure, A_t the nozzle throat area and C_F is the thrust coefficient. The thrust coefficient, which characterizes the thrust amplification due to finite ε (over the throat area alone), is a function of the specific heat ratio, nozzle exit-plane area ratio and the ratio of combustion pressure to ambient pressure. Thrust coefficients were conveniently obtained from the C_F plots given by Sutton[9] for $\gamma = 1.2$, corresponding to the N_2O_4/MMH case.

At an altitude of 5 Km, operation of the R-4D motor at a thrust of 500 lbs. requires a combustion-chamber pressure of 550 psia. Motor operation at this combustion pressure will be adopted for all ambient altitude conditions. As the ambient chamber pressure is reduced for the higher altitude experiments, C_F increases with a resulting increase in rocket motor thrust. However, as indicated in the table below, the effect is considered to be sufficiently small so as not to warrant the calculation of small changes in fuel and oxidizer charge tube pressures to effect small corrective adjustments in propellant mass flows for each altitude condition.

Combustion Chamber Pressure (psia)	$p_c/p_{amb.}$	Altitude (Km)	Thrust (lbs.) $\varepsilon = 6.8$	$\varepsilon = 12$
550	75	5	500	495
↓	285	15	527	540
↓	∞		535	555

4.2 Gaseous Simulation

Rocket performance calculations were made for the $N_2O_4/N_2H_3(CH_3)$ liquid propellant combination for a range of O/F_{Liquid} from 0.5 to 3.0. The calculations were performed using the computer program of Gordon and McBride.[10] The rocket specific impulse is shown in Figure 10 as a function of the liquid propellant O/F for the case of equilibrium and frozen composition during the nozzle expansion. The data correspond to the nozzle exit-plane area ratios to be employed in the experiments.

The $O/F_{Liquid} = 1.6$ mixture represents the operational case for this liquid bipropellant as it corresponds to equal tank sizes (equal volume of propellants). The initial gaseous propellant experiments in the altitude chamber will simulate this particular O/F_{Liquid} mixture ratio. The simulation will employ a mixture of O_2 and N_2 as the oxidizer and a mixture of H_2 and C_2H_4 for the fuel. The simulation requirements were obtained as follows:

	N_2O_4	+	$N_2H_3(CH_3)$	$(O/F_{Liquid} = 1.6)$
Weight (gm)	40		25	assumed for calculation
Molec. wt. (MW)	92.014		46.073	
moles	.4347		.5426	

$\therefore$ gm atoms $0_{ox} = \dfrac{64 \times 40}{92.014}$ gm atoms $N_F = 15.201$

$= 27.822$ gm atoms $C_F = 6.517$

gm atoms $N_{ox} = 12.178$ gm atoms $H_F = 3.282$

Total gm atoms

$N = 27.379$
$O = 27.822$
$C = 6.517$
$H = 3.282$

whence

$$N : O : C : H = 1 : 1.016 : 0.238 : 0.1175$$

Then for gaseous simulation,

$$wN_2 + x0_2 + y\, C_2H_4 + z\, H_2 = 27.379 \text{ gm N}$$
$$+ 27.822 \text{ gm O} + 6.517 \text{ gm C} + 3.282 \text{ gm H}$$

Solving for w, x, y and z in moles shows

$$.977\ N_2 + .869\ O_2 + .271\ C_2H_4 + 1.085\ H_2$$

$$= .435\ N_2O_4 + .543\ N_2H_3\ (CH_3)$$

For the oxidizer, therefore, total moles = 1.847 which, in terms of mole fractions X_j, is

$$0.52 N_2 + 0.48\ O_2$$

For the fuel, total moles = 1.3565, or

$$0.8\ H_2 + 0.2 C_2H_4$$

From the total gram atoms, it is seen that for the $O/F_{Liquid} = 1.6$ mixture, the equivalent gaseous simulation corresponds to $O/F_{Gaseous} = 5.7$.

The separate fuel and oxidizer gases, premixed in accordance with the above mole fraction distributions, have been received from the vendor.

Rocket performance calculations were made for the mixture of gaseous oxidizer ($0.52\ N_2 + 0.48\ 0_2$) and gaseous fuel ($0.80\ H_2 + 0.20\ C_2H_4$) at an $O/F_{Gas} = 5.7$ (by weight). The combustion chamber and exit-plane conditions are compared with those for the liquid propellant ($O/F_{Liquid} = 1.6$) in Table 1. The agreement is seen to be quite satisfactory. The corresponding specific impulse values for the gaseous propellants have also been included in Figure 10 for comparison (▲ symbols).

4.3 Gasdynamics of the Oxidizer and Fuel Systems

The oxidizer and fuel gases are loaded separately into coiled lengths of charge tube which serve as the propellant supply reservoirs. On motor initiation, a valve in each charge tube is opened and the required propellant mass flow rates are established by means of choked-flow venturis. The propellant gases then pass through the injector into the combustion chamber where combustion is initiated by means of a spark. The spark and separate valve operations are controlled by means of individual time delays so as to ensure the desired event synchronization.

The singular advantage of this system has been the fact that reliable and repeatable motor operation has been achieved without the use of a diaphragm in the nozzle throat region. The required altitude-chamber ambient conditions therefore only need to be re-established for the next run, without the necessity of returning the chamber to atmospheric pressure for diaphragm replacement purposes. The nozzle flow is terminated after about 50 msec by closure of the charge tube valves.

The rocket nozzle mass flux is given, in English units, by

$$\dot{w} = p_c A_t g \frac{f(\gamma)}{\sqrt{g\gamma RT}} \text{ lbm/sec}$$

$$= 0.1443\, p_c A_t \sqrt{\frac{MW}{\gamma T}}\; f(\gamma) \text{ lbm/sec}$$

where $f(\gamma) = \sqrt{\left(\frac{2}{\gamma+1}\right)^{\gamma+1/\gamma-1}}$,

p_c is the combustion pressure (psia), A_t is the nozzle throat area (in^2), MW is the molecular weight (gm $mole^{-1}$), R the gas constant ($\frac{1545.3}{MW}$ ft. lb/$_{lb.\,°R}$) and T is the combustion temperature (°R).

For the gaseous propellant combination,

Oxidizer: $.52N_2 + .48O_2$

Fuel: $.80H_2 + .20\ C_2H_4$

and the chamber conditions for 500 lb. thrust shown in Table 1,

$$\dot{w} = \dot{w}_{ox} + \dot{w}_F = 1.8283 \text{ lbm/sec}$$

$$r = \frac{\dot{w}_{ox}}{\dot{w}_F} = 5.7$$

$$\therefore \quad \dot{w}_{ox} = \frac{r}{r+1} \quad \dot{w} = 1.555 \text{ lbm/sec}$$

$$\dot{w}_F = \frac{1}{r+1} \, \dot{w} = 0.2729 \text{ lb/sec}$$

Noting the specific heat data for the propellant species as

$C_{p_{N_2}}$	= .2477 BTU/lb°R	= 6.939 BTU/mole°R
O_2	= .2178	6.970
H_2	= 3.389	6.832
C_2H_4	= .359	10.071

the following pertinent propellant data are evaluated:

<u>Oxidizer</u> $\quad MW_{ox} \quad = \quad \sum X_j MW_j = 29.927$ gm/mole

$$R_{ox} = \frac{1545.3}{MW} = 51.636 \text{ ft.lb/lb°R}$$

$$C_{p_{ox}} = \sum X_j \, C_{p\,j\,molar} = 0.2324 \text{ BTU/lb°R}$$

$$\gamma_{ox} = \frac{C_p}{C_p - R/J} = 1.4$$

sound speed $a_{ox} = \sqrt{\gamma g R T} = 1110$ ft/sec.

<u>Fuel</u> $\quad MW_F \quad = \; 7.223$ gm/mole

R_F = 213.936 ft. lb./lb°R

Cp_F = 1.036 BTU/lb°R

γ_F = 1.361

a_F = 2228 ft/sec.

4.4 Injector Requirements

The injector employed in the previous ARPA high-altitude plume program[4] was of a triplet design with inner and outer spray rings which caused the oxidizer to impinge on a central fuel spray. The suitability of this injector was examined for use in the present program.

The pressure ratio across the injector was assumed to be 1.3, a pressure drop which has been determined on the basis of considerable previous experience with this injector, i.e.

$$p_{inj} = 1.3\, p_c = 715 \text{ psia}$$

The injection hole-size requirements to accommodate the required oxidizer and fuel mass flows for the present experiments were then determined.

The mass flow through the injector is given by (English Units),

$$\dot{w}_{ox}, \dot{w}_F = \frac{A_{inj}\, p_{inj}}{R} \sqrt{2gJ} \sqrt{\frac{C_p}{T}} \sqrt{\left(\frac{p_c}{p_{inj}}\right)^{2/\gamma} - \left(\frac{p_c}{p_{inj}}\right)^{\gamma+1/\gamma}} \text{ lbm/sec}$$

where $\dot{w}_{ox}$ = 1.555 lb/sec

$\dot{w}_F$ = 0.2729 lb/sec

and J is the mechanical equivalent of heat (778 ft. lb/BTU). The other gas-dynamic parameters for the cold (T = 530°R) oxidizer and fuel mixtures have been previously evaluated above. The requirements for injector-hole area show

$$A_{inj_{ox}} = .1065 \text{ in}^2$$

$$A_{inj_{fuel}} = .0386 \text{ in}^2$$

The present injector has 32 holes for injection of the oxidizer, whence, assuming a reasonable discharge coefficient, $C_D = 0.7$,

$$d_{inj_{ox}} = 0.077 \text{ in.}$$

Also, fuel injection is effected through 16 holes, which, for $C_D = 0.7$ requires

$$d_{inj_{fuel}} = 0.066 \text{ in.}$$

For the present injector,

$$d_{inj_{ox}} = .072 \text{ in.}$$

$$d_{inj_F} = .060 \text{ in.}$$

These results indicate that the existing injector will be suitable.

4.5 Propellant Venturis

The oxidizer and fuel mass flows to establish the desired combustion pressure level and O/F_{Liquid} simulation are determined by the parameter product of propellant charge tube supply pressure, p_{CT}, and the venturi throat area, A_v. The lower the charge tube pressure, the larger the venturi size required. However, although it is desirable that the initial charge tube pressures be no higher than is necessary, additional considerations are imposed on these parameters.

When the valve in the charge tube is opened, an expansion fan is propagated along the length of charge tube. Strictly, the pressure, p_v, at the inlet to the venturi, along with the venturi throat size, establishes the mass flow. The venturi inlet pressure is therefore that corresponding to the reduced level of charge-tube pressure across the expansion fan. For a given γ gas, the ratio β $(= p_v/p_{CT})$ is a function of the ratio of venturi throat area to charge tube area, A_v/A_{CT}, and β decreases as A_v/A_{CT} increases. For a given A_v, therefore, the appropriate β value must be included in the determination of the required charge-tube pressure.

The venturis are designed with a 4° (1/2 ∢) expansion after the throat to provide some diffuser pressure recovery. The pressure ratio across the venturi is then of order 1.4. Overall, therefore, for the venturi and injector,

$$p_v/p_c \approx 1.3 \times 1.4 \approx 2$$

whence $p_v = 1100$ psia.

From the rocket nozzle mass-flow relation cited earlier, the oxidizer and fuel mass flows may be written as

$$\dot{w}_{ox}, \dot{w}_F = 0.1443\, p_v A_v \sqrt{\frac{MW}{\gamma T}} \text{ lb/sec.}$$

$$\therefore \dot{w}_{ox} = .0235\, p_v A_v = 1.555 \text{ lb/sec.}$$

and $$\dot{w}_F = .0114\, p_v A_v = 0.2729 \text{ lb/sec.}$$

Therefore, for the oxidizer

$$p_v A_v = 66.244$$

$$A_v = .0602 \text{ in}^2$$

$$d_v = .277 \text{ in}$$

Also, charge tube diameter = 0.865 in

$$\therefore \quad A_v/A_{CT} = .1024$$

$$\beta = .923 \; (\gamma = 1.4)$$

and $\qquad p_{CT} = 11.92$ psia

For the fuel,

$$p_v A_v = 23.923$$

$$A_v = .0217 \text{ in}^2$$

$$d_v = .166 \text{ in}$$

Then $\qquad A_v/A_{CT} = .0369$

$$\beta = .972 \; (\gamma = 1.36)$$

and $\qquad p_{CT} = 1132$ psia.

The above calculations were similarly performed to encompass a range of charge-tube pressures (i.e. fuel and oxidizer mass flows) for the above (now fabricated) venturi sizes. Since the respective oxidizer and fuel A_v is fixed, the A_v/A_{CT} values and therefore the β values are constant and $\dot{w}_{ox}$, $\dot{w}_F$ may be scaled directly, i.e.

$$\dot{w}_{ox} = .0235 \, (A_v \beta \, p_{CT})_{ox} = .001306 \, p_{CT_{ox}}$$

$$\dot{w}_F = .0114 \, (A_v \beta \, p_{CT})_F = .0002405 \, p_{CT_F}$$

Therefore, for $O/F_{Gas.} = 5.7$,

$$p_{CT_F} = \frac{\dot{w}_{ox}}{5.7 \times .0002405 \text{ psia}}$$

The above calculations are summarized in Figure 11 and provide for adjustment of the combustion-chamber pressure level during the experiments, if required, while maintaining the desired propellant O/F_{Gas}. For a particular oxidizer charge tube pressure, the upper curve indicates the oxidizer mass flow rate, $\dot{w}_{ox}$, and the lower curve indicates the fuel charge-tube pressure required to maintain $\dot{w}_{ox}/\dot{w}_F = 5.7$ at a combustion pressure level $\sim p_{CT}/2$.

4.6 Duration of Steady Combustion Pressure

Following the opening of the valves in the charge tubes, the pressure at the inlet to the venturis will remain steady over the time interval required for the expansion fan to propagate to the end of the charge tube and return to the venturi. The arrival of the reflected rarefaction wave may then unchoke the venturis.

The oxidizer charge-tube length is about 25 feet. At the sound speed calculated previously for the oxidizer gas, the expansion wave transit time is about 45 msecs. The longer length (54 feet) of fuel charge tube, necessitated by the higher sound speed in the fuel mixture, results in a wave transit time of about 48 msecs. As noted previously, the charge tube valves are closed after about this time interval in order to conserve propellant gases.

4.7 Variation of O/F

The initial experiments will be performed at an $O/F_{Gaseous} = 5.7$ which provides good duplication of the N_2O_4/MMH propellants at $O/F_{Liquid} = 1.6$ (Table 1). It is intended, further, to investigate the effect of O/F variation in the experiments. The requirements for the simulation of variable O/F_{Liquid} were determined by calculations similar to those described previously for the $O/F_{Liquid} = 1.6$ mixture. Variation of O/F_{Liquid} is accomplished by adjustment of the mole fractions of N_2 and O_2 in the oxidizer mixture only, the fuel requirements remaining unchanged from the $0.80\ H_2 + 0.20\ C_2H_4$ mixture.

The N_2 - O_2 oxidizer mole fractions and the corresponding gaseous equivalent $O/F_{Gaseous}$ for simulation of N_2O_4/MMH at various O/F_{Liquid} are shown in Figure 12. It is seen that a comparatively small adjustment in the respective oxidizer X_j values is required to effect a change in O/F_{Liquid}. Careful control of the oxidizer-gas mole fractions in the mixing preparation is therefore necessary. For this reason, in ordering the gases from the vendor (Matheson) it was specified that the mixture preparations be held to within 1% by volume. A requested gas analysis which accompanied each mixture preparation (fuel and oxidizer) confirmed the required tolerance.

In order to enhance afterburning in the plume experiments performed in ambient air, the motor would be run at more fuel-rich conditions. It is noted, however, that each change in O/F requires a new oxidizer venturi. For example, the case N_2O_4/MMH at $O/F_{Liquid} = 1.0$ requires a gaseous oxidizer combination of 0.60 N_2 + 0.40 O_2 at a charge-tube pressure of 1155 psia, and an oxidizer venturi throat diameter, $d_{V_{ox}}$ = 0.212 in. The fuel mixture, pressure and venturi size are unchanged.

As required, 1% mixtures will be supplied by a vendor. The necessary oxidizer combinations could be obtained in situ, by the controlled addition of N_2 (or O_2) to the existing (.52 N_2 + .48 O_2) mixture, to establish the required partial pressures in the charge tube. However, mixing times required for volume homogeneity with this technique could adversely affect the run rate.

4.8 Altitude Chamber Purging

Following each motor run, the altitude chamber must be pumped down to reduce to negligible levels the absorption by the partial pressures of the CO_2 and H_2O gases in the chamber volume. The tank pressure is then returned (using air or N_2) to the equivalent altitude level required. Each rocket firing, of order 50 msecs duration, exhausts about

40 gms of combustion gases into the altitude chamber. The post-run partial pressures of CO_2 and H_2O in the tank volume ($\sim$ 2200 ft^3) are therefore about 20 μm Hg and 185 μm Hg respectively for any ambient altitude condition.

For 5 Km altitude, the ambient tank pressure is about 400 torr, at which the CO_2 and H_2O correspond to contaminant levels of 50 ppm and 460 ppm respectively. Therefore, reduction of the tank pressure to 1 torr and return to 400 torr with N_2 (or air) will reduce the CO_2 and H_2O concentrations to 0.1 and 1 ppm respectively.

The purge requirements are stricter at the higher altitude conditions. At 30 Km, for example, the ambient tank pressure is only 9 torr. If, in this case, the tank is pumped down to 100 μm Hg (a level quite rapidly achieved with the mechanical pumps on the altitude - chamber facility) and returned to 9 torr, the CO_2 and H_2O concentration levels are about 25 ppm and 230 ppm respectively. A repeat of the above purge to 100 μm Hg would then reduce these CO_2 and H_2O levels to about 0.3 ppm and 2.5 ppm.

SECTION 5

TEST MATRIX AND PROGRAM STATUS

5.1 Test Matrix

As mentioned in Section 1, the program plan involving the Ludwieg tube tests was altered to perform first the chemistry-oriented chamber tests with no external flow. Thus, the test configurations and parameters have been integrated to make them compatible and interchangeable. Under the present program the following test conditions and matrix are planned:

a) A 500 lb. thrust motor will be run, using the gaseous equivalent propellant formulations corresponding to N_2O_4/MMH, N_2O_4/N_2H_4 as well as H_2/O_2 and $H_2/O_2/CO$ to be tested later.

b) Conical nozzle configurations with expansion ratios of 6.85 and 13.7 will be used.

c) Two clustered-nozzle configurations are planned.

d) Tests will be conducted in the altitude range between 5 and 35 km.

e) Both air and baseline N_2 tests will be conducted to assess the role of O_2 in afterburning.

f) The O/F ratio will also be varied as a test parameter.

5.2 Current Status

Checkout firing tests with the rocket motor in its 500 lb. thrust configuration are underway. These tests serve to determine the precise synchronization of fuel and oxidizer injection and spark ignition times. All the diagnostic instrumentation discussed in Section 3 has been installed and will undergo final calibration after the checkout test series is completed. The initial tests will use the N_2O_4 / MMH propellant system, the $\varepsilon = 6.85$ nozzle, variable O/F ratio, and will proceed to the higher-altitude conditions from an initial altitude of 5 km.

A supply line has been installed to connect a Linde Air-supplied nitrogen farm to the altitude chamber. This nitrogen is rated at 300 ppm pure. Tests in air will utilize the appropriate addition and mixing of pure O_2 with the N_2.

It is planned to complete this test matrix in July, 1974. A final report on this phase of the low-altitude plume radiance study is scheduled for 15 August, 1974.

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TABLE I

COMPARISON OF REACTION PRODUCTS OF

LIQUID AND GASEOUS PROPELLANTS

	LIQUID N_2O_4/MMH $O/F_{Liq.} = 1.6$		GASEOUS N_2-O_2/H_2-C_2H_4 $O/F_{Gas.} = 5.7$	
	Combustion Chamber	$\epsilon = 6.65$	Combustion Chamber	$\epsilon = 6.70$
Pressure (psia)	550	11.0	550	11.0
Temperature (°K)	3121	1560	3139	1588
Molec. wt.	20.35	20.65	20.57	20.90
Specific heat ratio	1.177	1.252	1.172	1.249
Species Mole fractions, X_j				
CO	.1351	.1100	.1322	.1073
CO_2	.0348	.0623	.0377	.0654
H_2	.1706	.1994	.1587	.1861
H_2O	.3260	.3177	.3375	.3322
N_2	.3051	.3105	.3031	.3090
H	.0152		.0154	

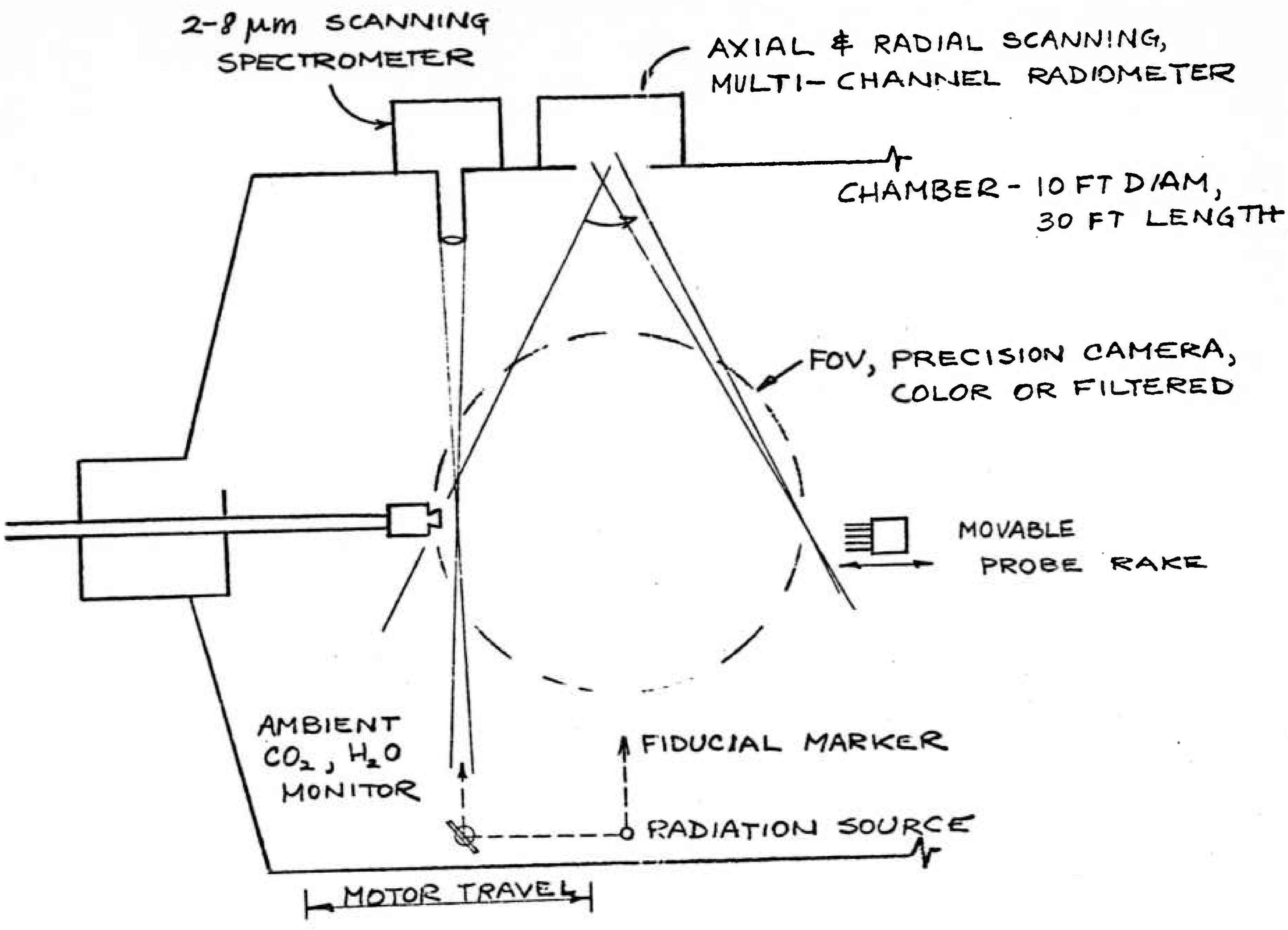


FIGURE 1 EXPERIMENTAL CONFIGURATION, DARPA PLUME STUDIES

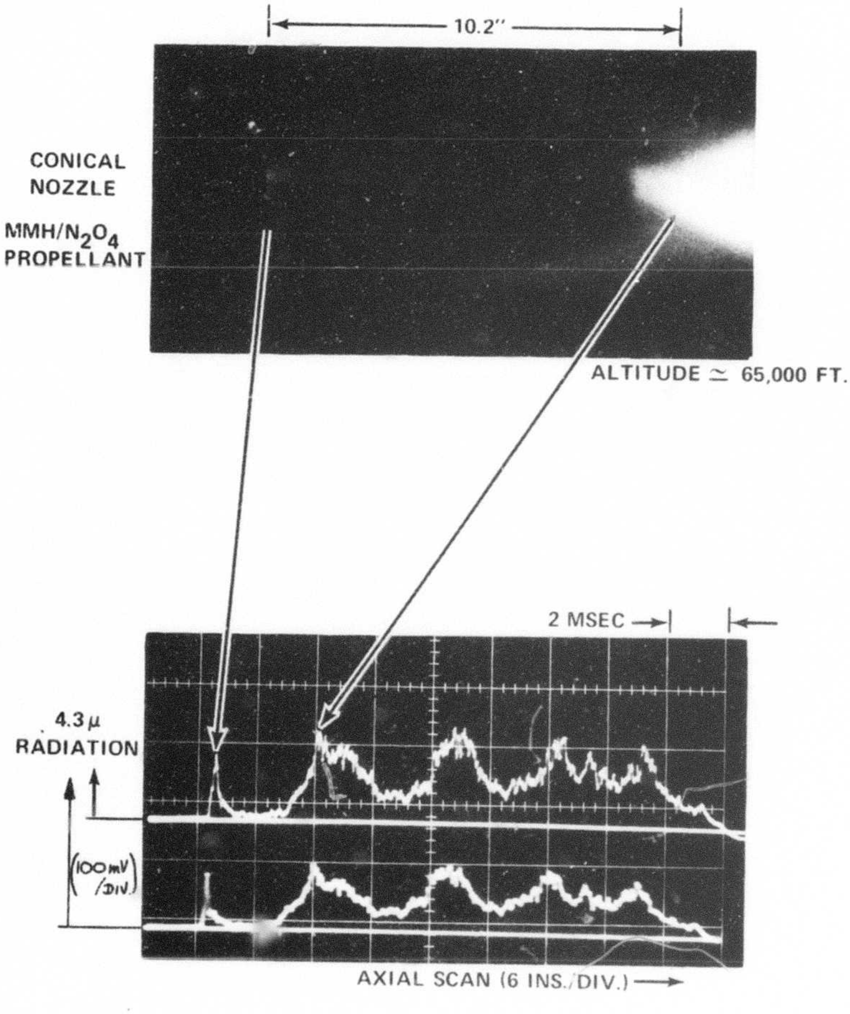


Figure 2 AXIAL RADIOMETER SCAN (4.3 μm) OF A LOW-ALTITUDE PLUME N_2O_4/MMH PROPELLANT, CONICAL NOZZLE

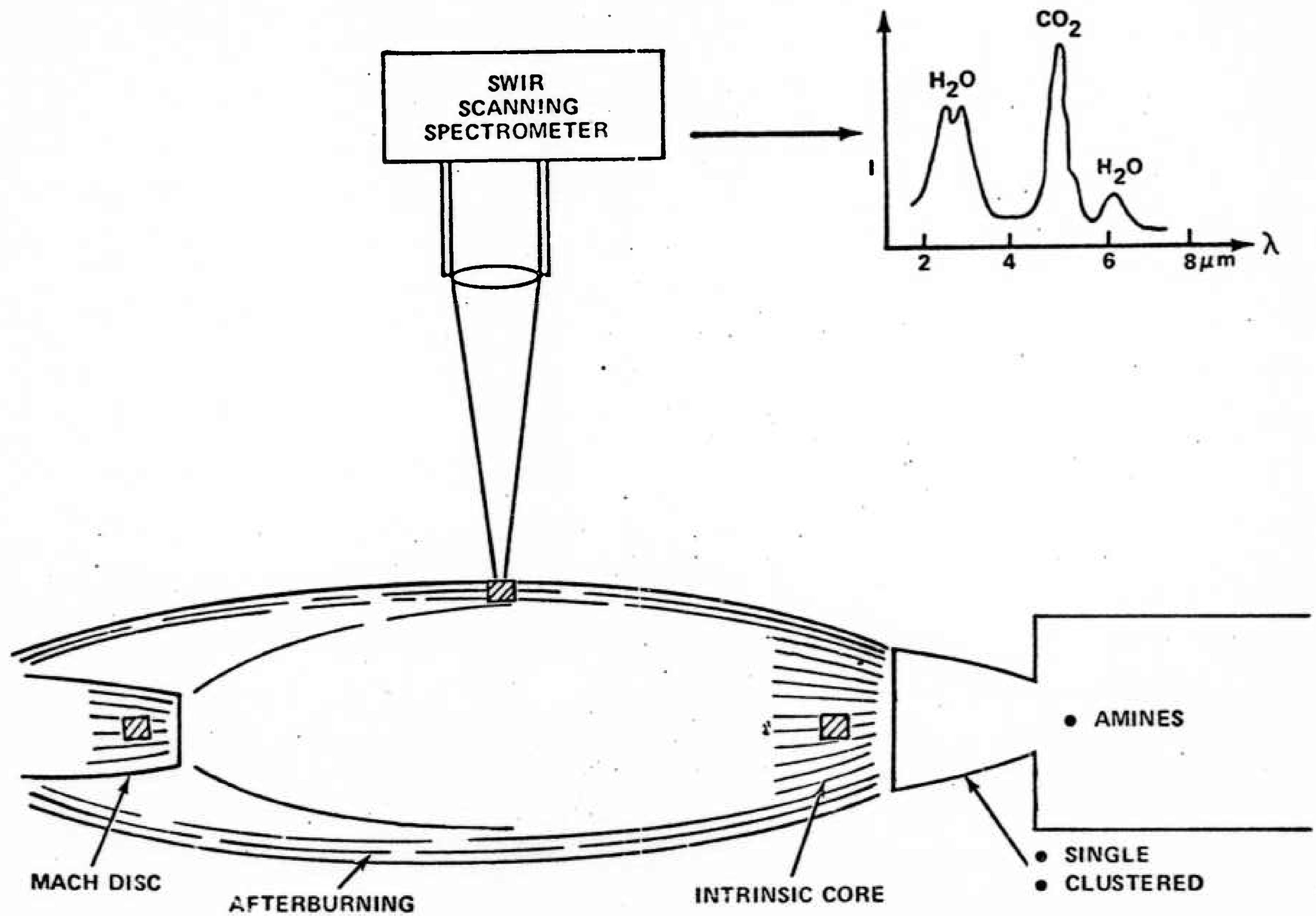


FIGURE 3 SCHEMATIC OF TEST SETUP WITH SWIR SCANNING SPECTROMETER

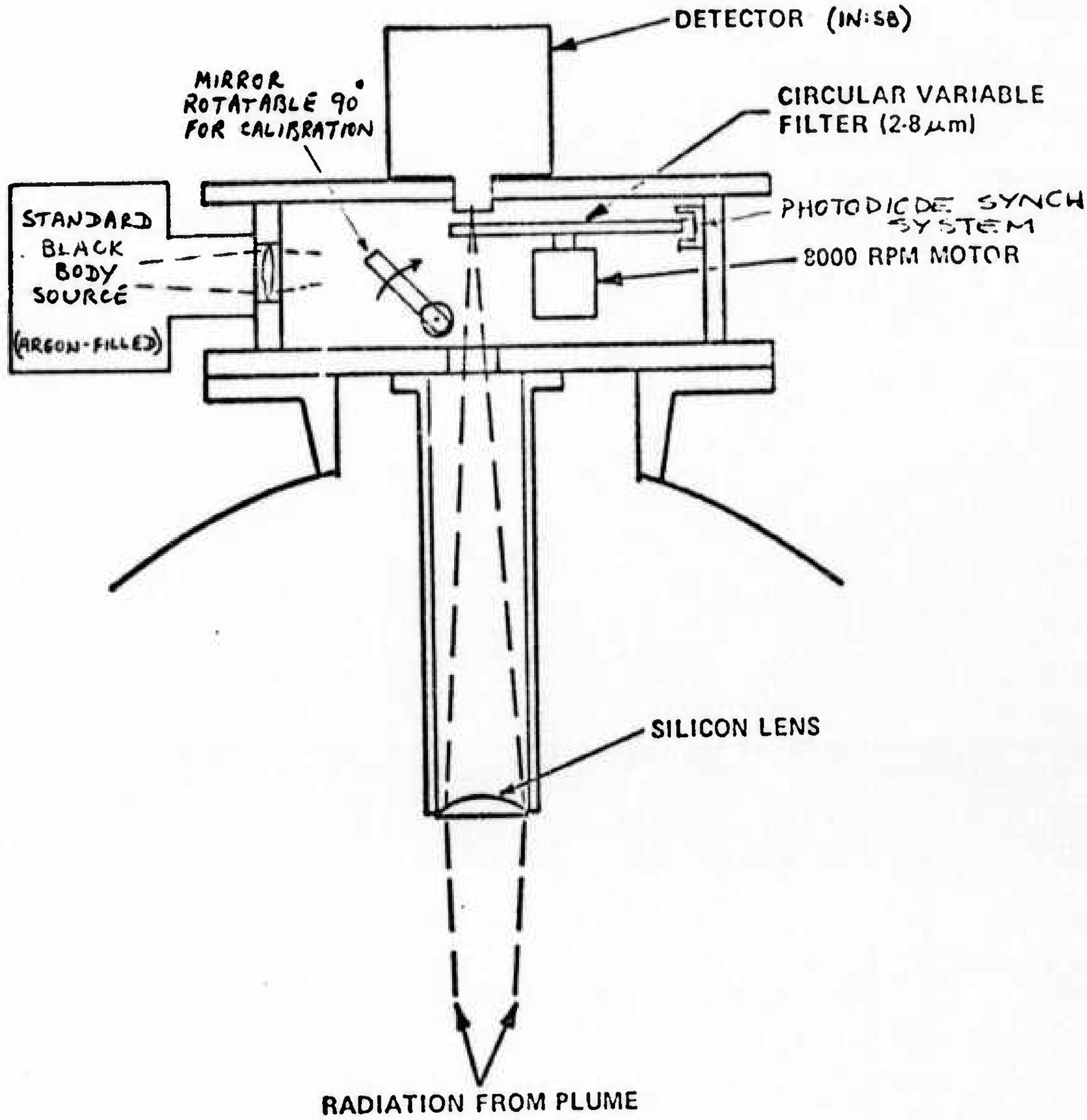


Figure 4 SCHEMATIC OF SWIR RAPID-SCAN SPECTROMETER

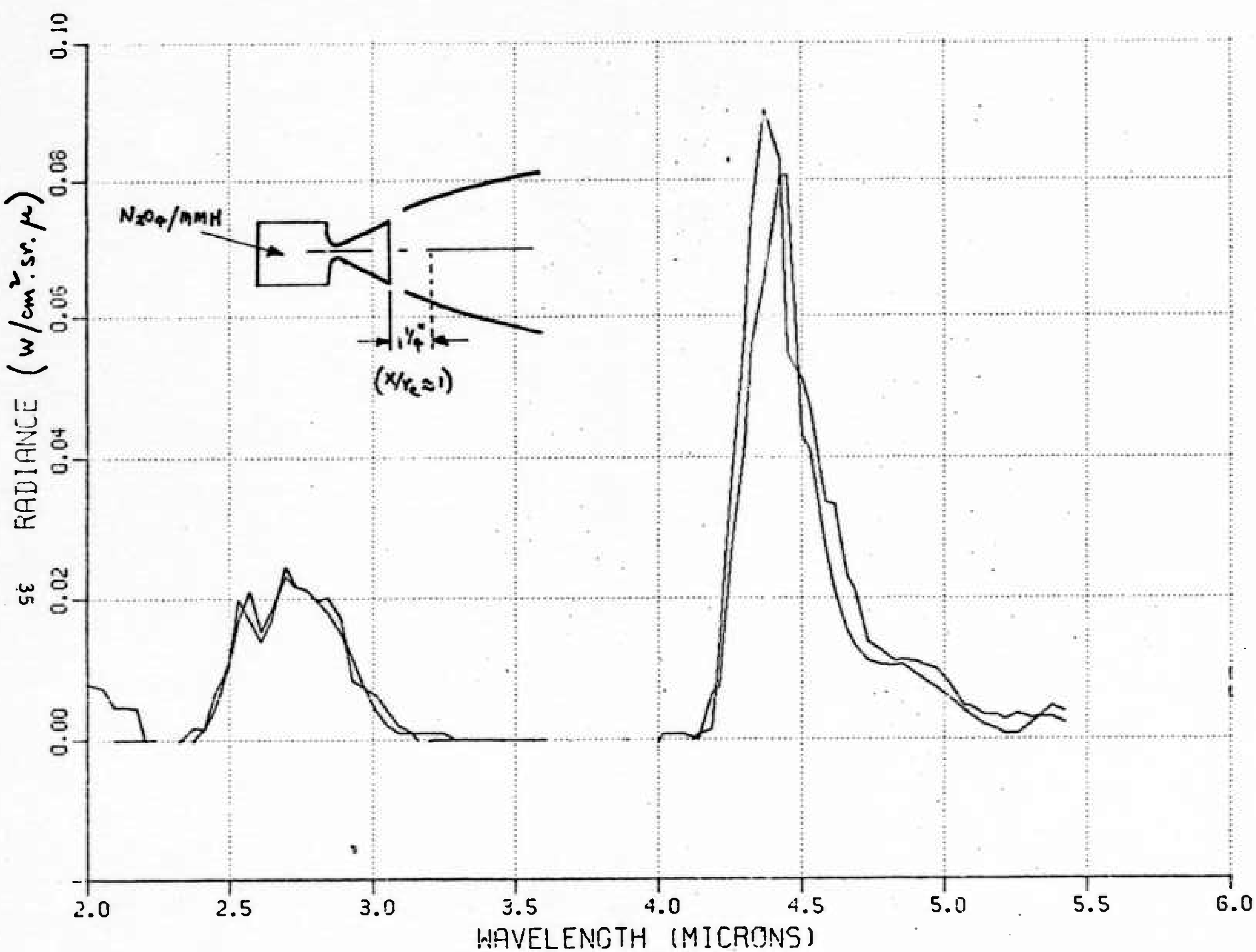


FIGURE 5 SWIR SPECTRAL SCANS - N_2O_4/MMH PROPELLANTS, CONICAL NOZZLE (X = 1 1/4 IN.)

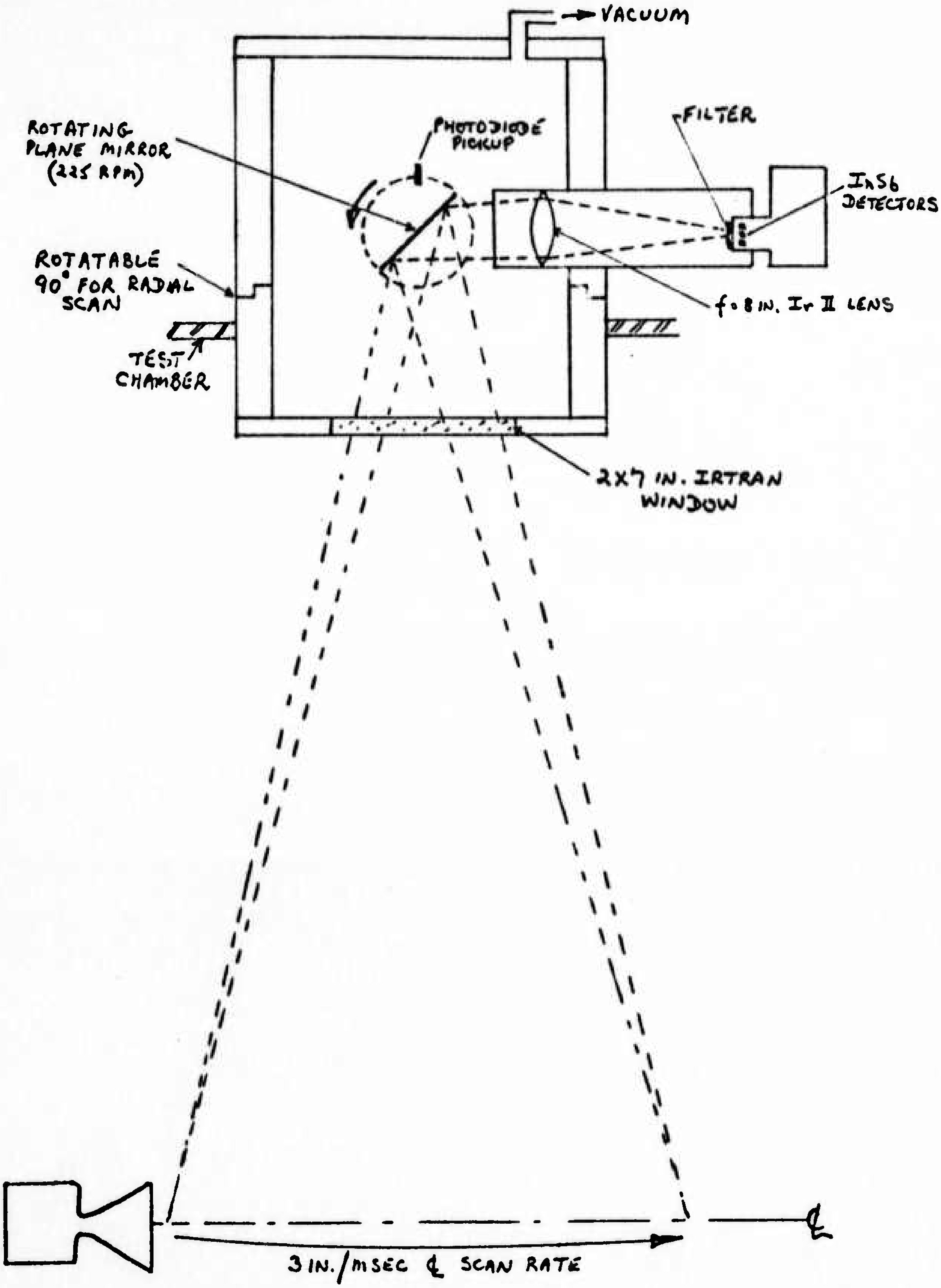


FIGURE 6 SPATIALLY SCANNING RADIOMETER

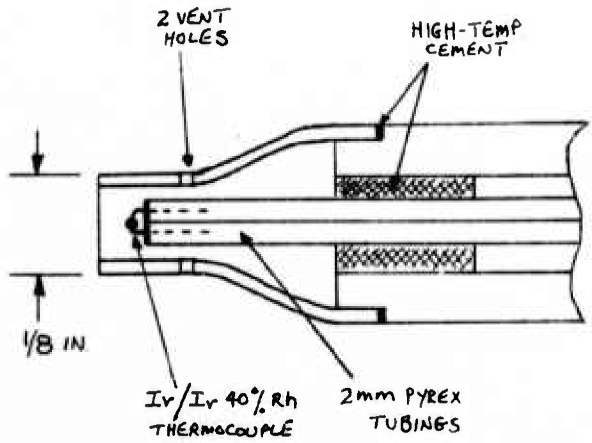


FIGURE 7 SCHEMATIC OF TOTAL-TEMPERATURE PROBE

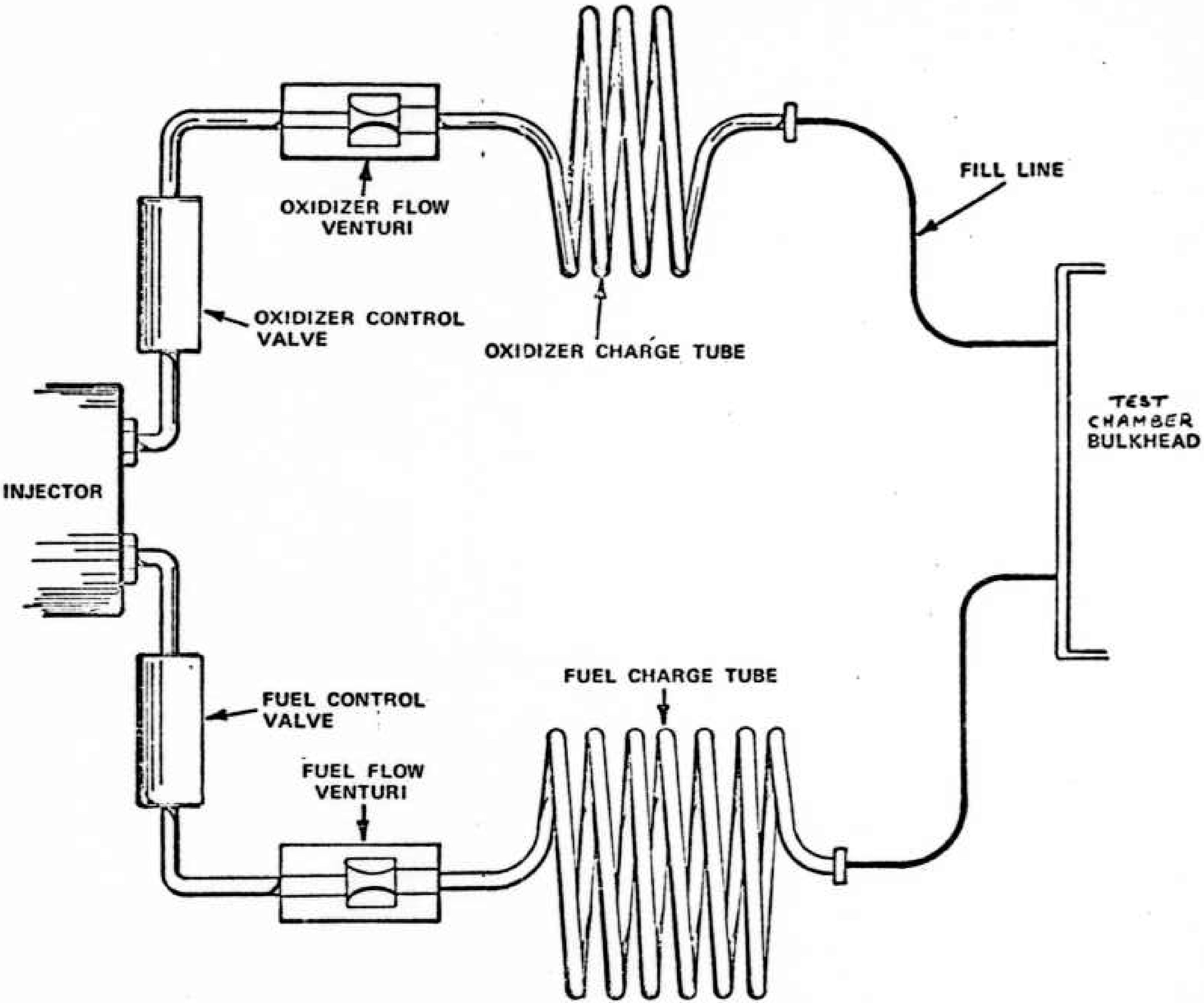
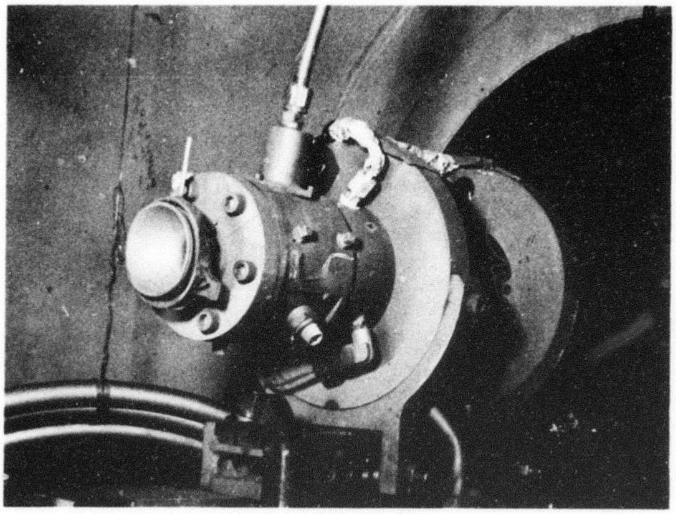
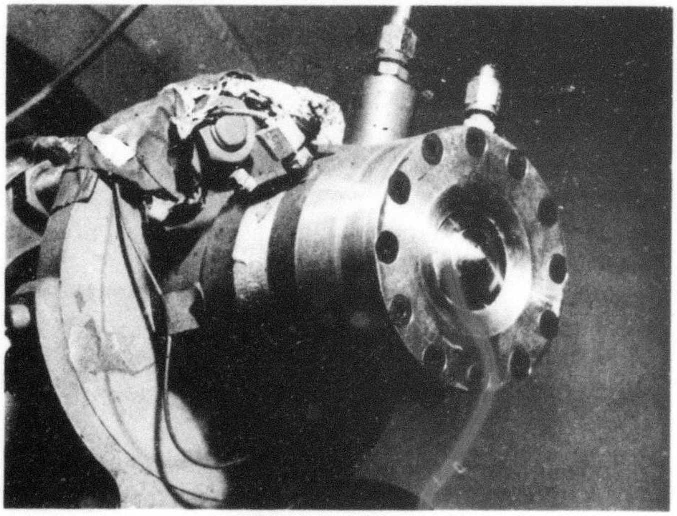


Figure 8 GASEOUS PROPELLANT SYSTEM SCHEMATIC



CONICAL NOZZLE



PLUG NOZZLE

Figure 9 N_2O_4/AMINE FUEL TEST MOTORS

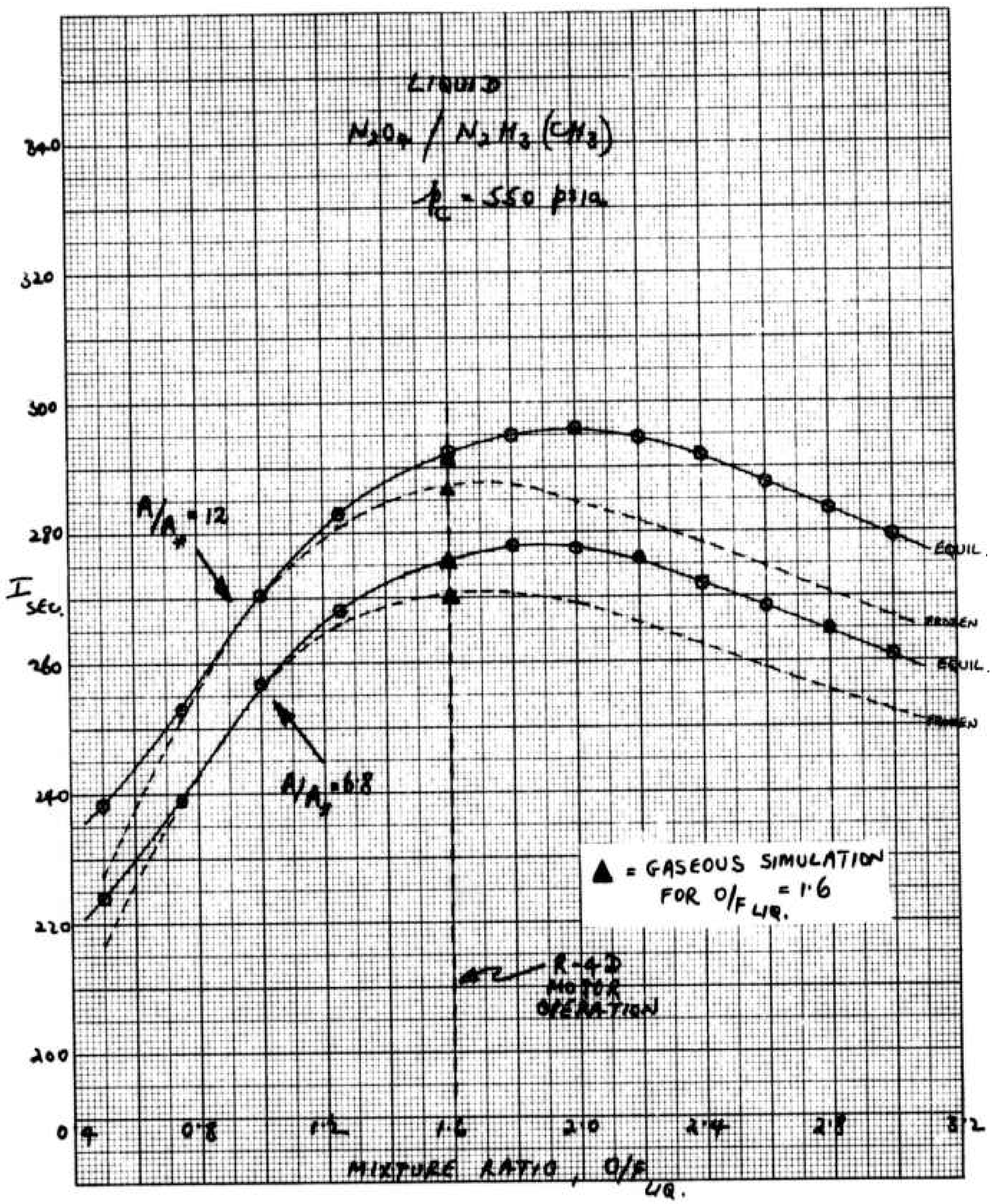


FIGURE 10 SPECIFIC IMPULSE OF R-4D MOTOR AS A FUNCTION OF LIQUID PROPELLANT MIXTURE RATIO

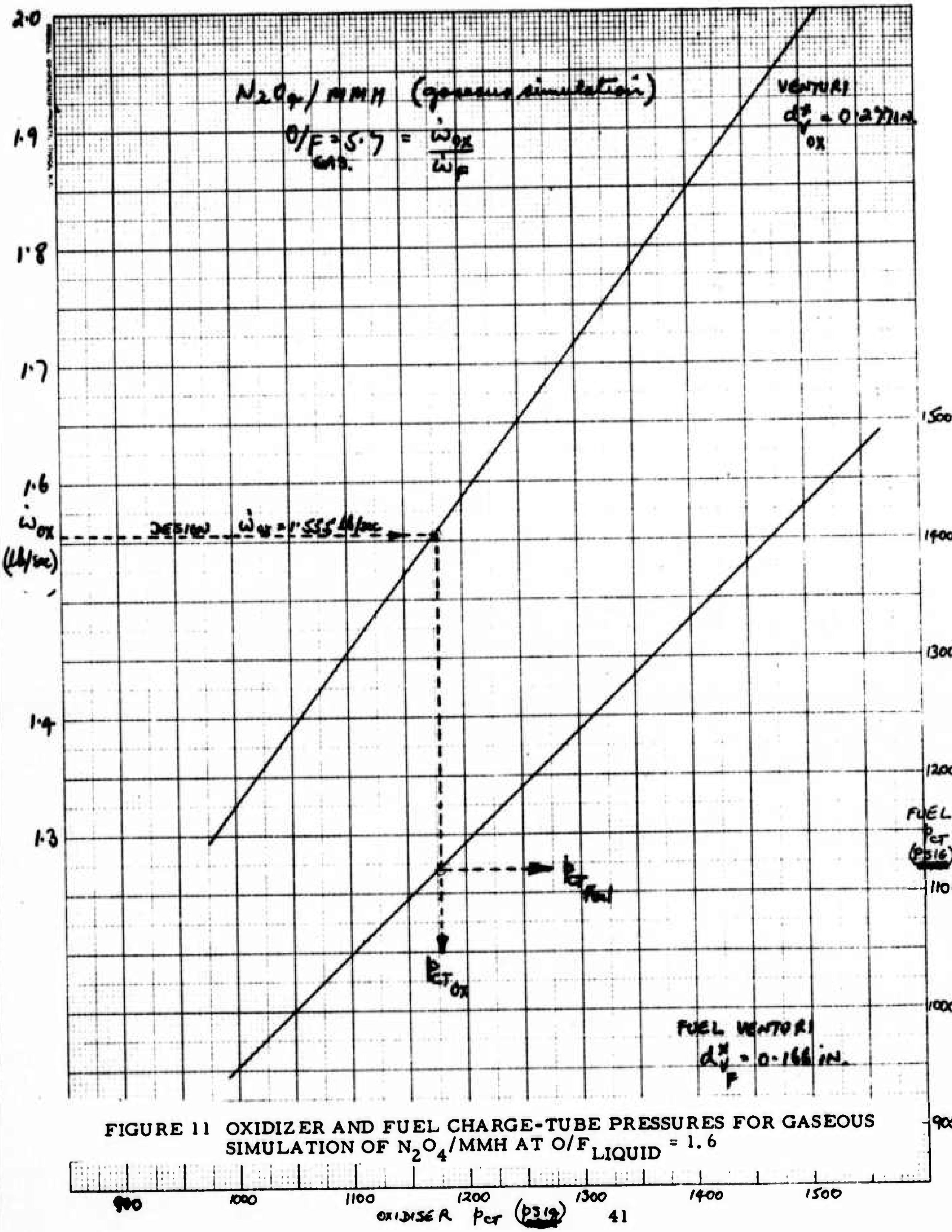


FIGURE 11 OXIDIZER AND FUEL CHARGE-TUBE PRESSURES FOR GASEOUS SIMULATION OF N_2O_4/MMH AT $O/F_{LIQUID} = 1.6$

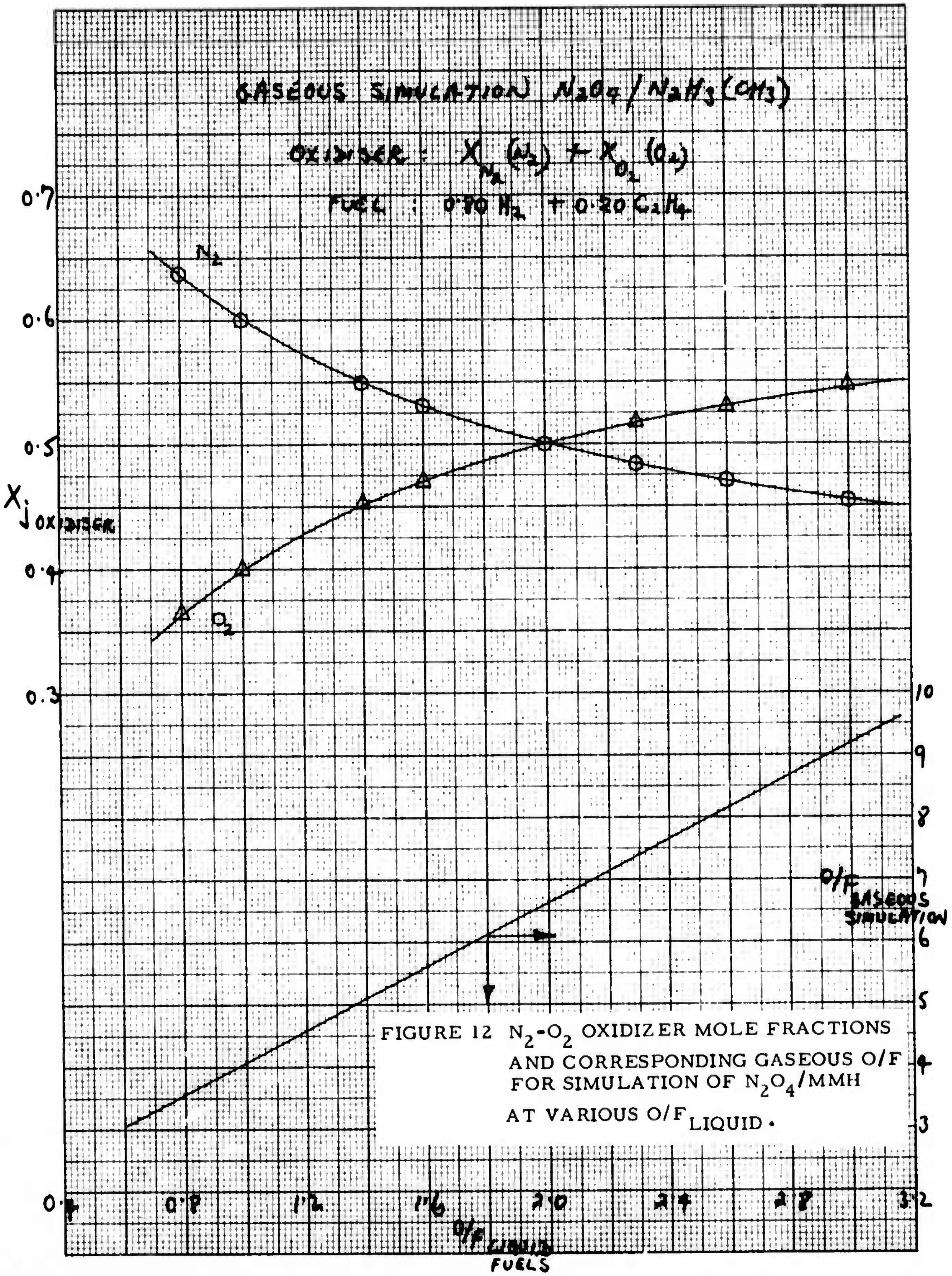


FIGURE 12 N_2-O_2 OXIDIZER MOLE FRACTIONS AND CORRESPONDING GASEOUS O/F FOR SIMULATION OF N_2O_4/MMH AT VARIOUS O/F_{LIQUID}.